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THE SMALL ASTRONOMY SATELLITE (SAS) PROGRAM

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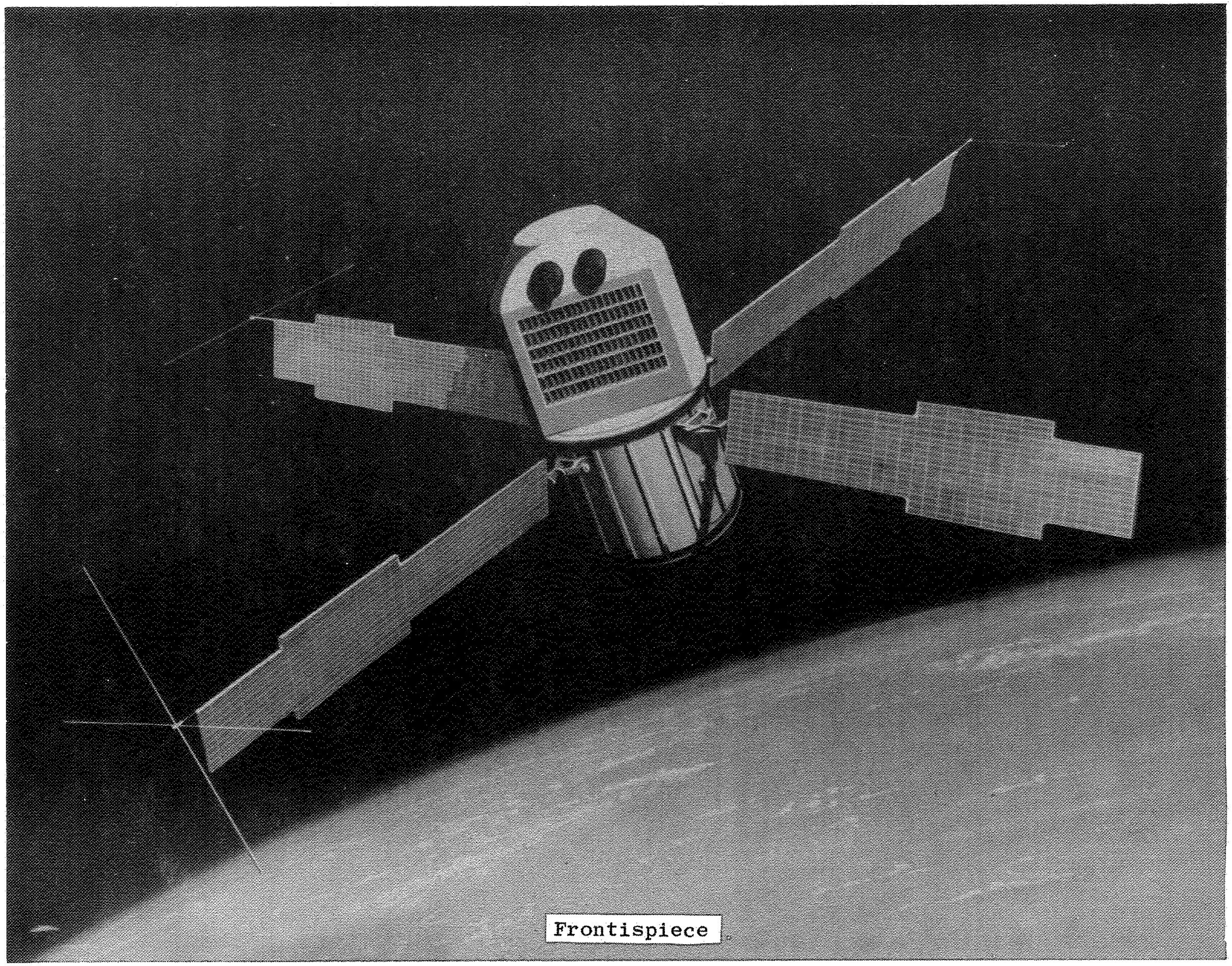
By
Marjorie R. Townsend

September 1968

GODDARD SPACE FLIGHT CENTER
Greenbelt, Maryland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THE SMALL ASTRONOMY SATELLITE (SAS) PROGRAM

Marjorie R. Townsend

ABSTRACT

This report provides a description of the Small Astronomy Satellite (SAS). As one of NASA's newest Explorer class programs, it has the capability of providing at relatively low cost, much basic, previously unavailable information concerning low and high energy radiation emanating from sources both inside and outside of our galaxy. Sufficient information is given concerning its volume, weight, power, commands, telemetry and control system to ascertain its ability to perform specific astronomy experiments. Project reliability requirements and environmental test conditions are included in the appendix.

CONTENTS

ILLUSTRATIONS

THE SMALL ASTRONOMY SATELLITE (SAS) PROGRAM

INTRODUCTION

The objective of the SAS program is to study the celestial sphere above the earth's atmosphere and to search for sources radiating in the X-ray, Gamma-ray, ultra-violet, visible, and infrared regions of the spectrum both inside and outside of our galaxy. Surveys performed from such a small, Scout launched satellite (Frontispiece) will provide valuable data on the position, strength, and time variation of sources radiating in these regions of the spectrum and lead to the selection of the more interesting ones for detailed study by later missions on this or more sophisticated spacecraft It is anticipated that one of these satellites will be launched each year, with an estimated lifetime of a year, unless limited by the experiment sensors.

The planned orbital characteristics of the SAS series are a nominally circular 300 nautical mile (555 kilometer) equatorial orbit. The equatorial orbit is highly desirable for an astronomy mission because the satellite will miss the South Atlantic anomaly where the radiation belts come far down into the earth's atmosphere (Figure 1). This will avoid degradation of the spacecraft operation and maintain at a minimum the background count which can affect the data returned from several different types of sensors applicable to satellite astronomy. This orbit, with an inclination of 2.9 degrees, can be achieved by launching from the San Marco platform located three miles east of Kenya in the Indian Ocean. The maximum payload capability under these conditions is 333 pounds.

BASIC SPACECRAFT DESIGN

The satellite is designed to have a clean interface between the basic spacecraft structure and the experiment payload (Figure 2). This is intended to minimize costs for follow-on programs. The basic spacecraft is a cylinder approximately 22 inches in diameter and 20 inches high, weighing about 180 pounds. The lightweight cylindrical shell acts as the primary support for the experiment. Four solar paddles, hinged to the outer shell, provide raw power to the spacecraft and experiment.

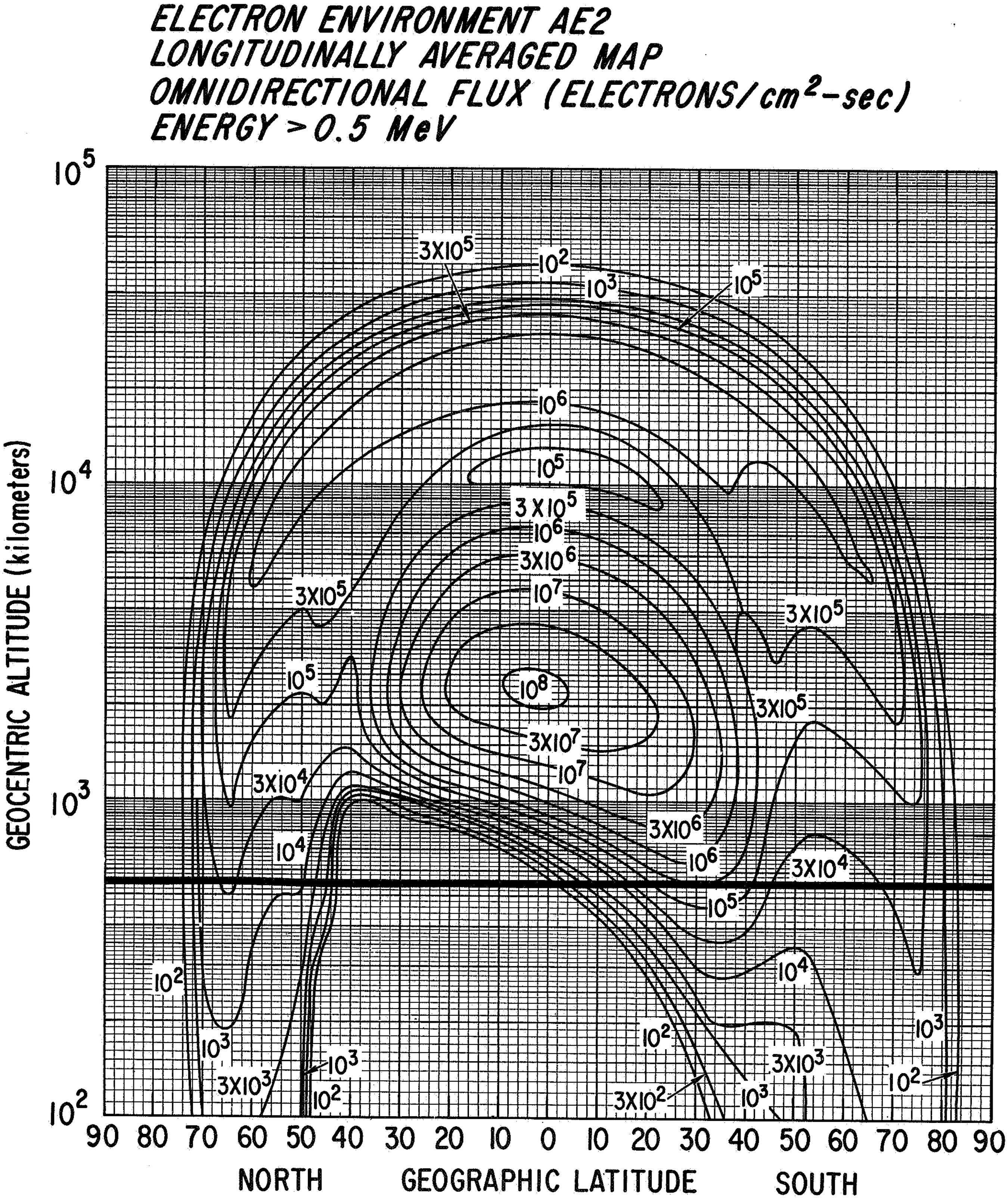

ELECTRON ENVIRONMENT AE2
LONGITUDINALLY AVERAGED MAP
OMNIDIRECTIONAL FLUX (ELECTRONS/cm²-sec)
ENERGY > 0.5 MeV
GEOCENTRIC ALTITUDE (kilometers)
10⁵
10⁴
10³
10²
90 80 70 60 50 40 30 20 10 0 10 20 30 40 50 60 70 80 90
NORTH
GEOGRAPHIC LATITUDE
SOUTH
10⁸
3X10⁷
10⁷
3X10⁶
10⁶
3X10⁵
10⁵
3X10⁴
10⁴
3X10³
10³
3X10²
10²

Figure 1

SAS
IN
LAUNCH CONFIGURATION
ON SCOUT
WITH
STANDARD SHROUD

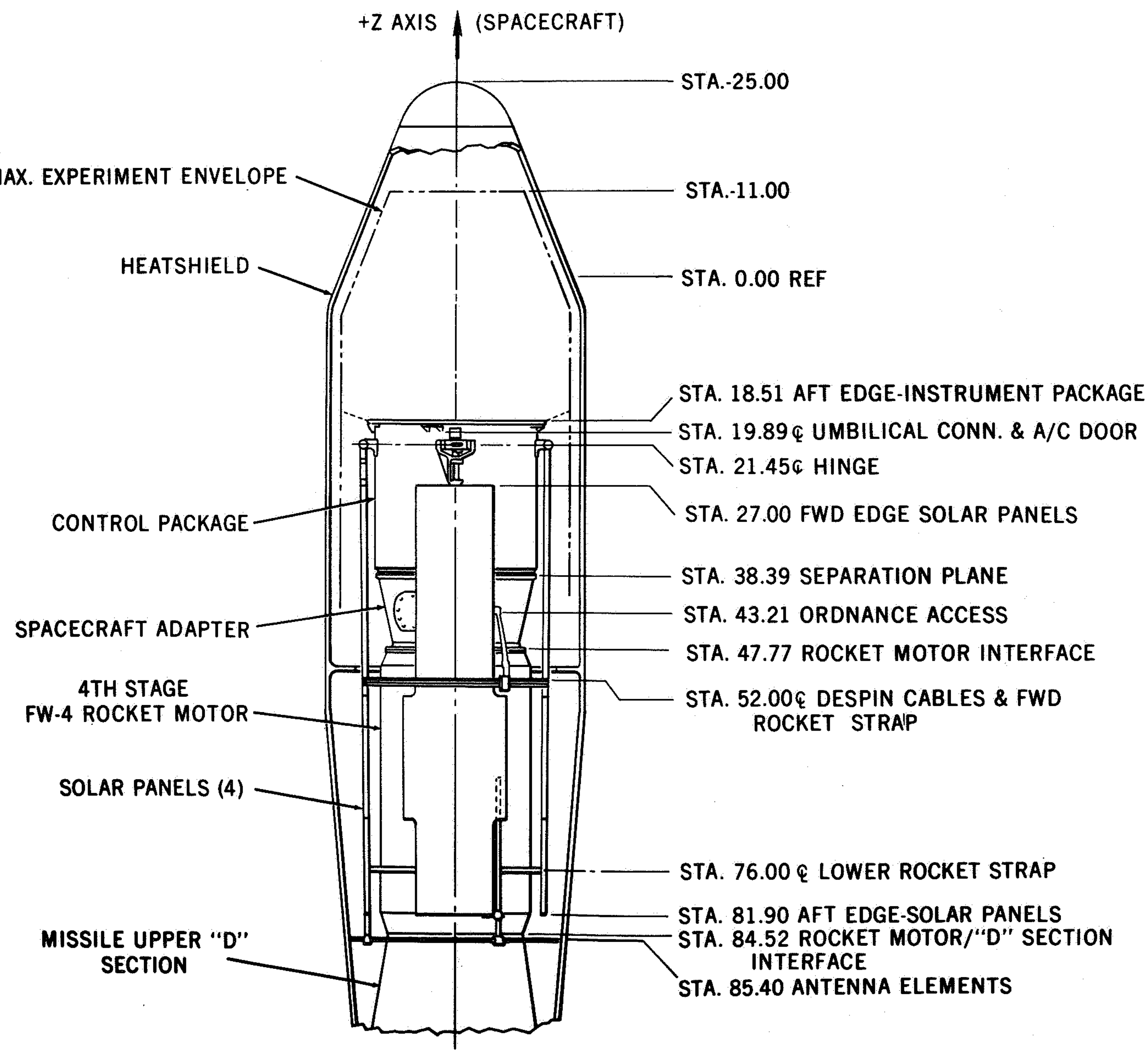


Figure 2

At the tips of the paddles are the command and telemetry antennas. Inside the shell (Figure 3), and thermally isolated from it, is a honeycomb deck which is itself a good thermal conductor. On it are the systems to provide the basic spacecraft functions: a rechargeable battery with its charge control and regulator systems; command receivers and decoders; a 1000 bit per second telemetry system with one orbit storage capability on tape; and a magnetically torqued commandable control system which can point the spacecraft stably to any point in the sky. The electronic circuitry is packaged in "books" mounted with the nutation damper on the underside of the honeycomb deck. The battery, tape recorder, transmitter, command receivers, and rotor are mounted on the top side.

One of the design requirements for SAS was to be able to point in any orientation, thus complicating the thermal design as well as that of the power system. The surface of the cylinder is the only part of the spacecraft portion which radiates the internal heatloads to space. Since the experiment is isolated from the spacecraft section with a high thermal resistance, we will consider the spacecraft section independently of the experiment section. The most critical items are the tape recorder and the battery, which are co-located for efficient thermal design. The judicious use of passive thermal coatings and multilayer insulation, with a small amount of active thermal control in the form of heaters, permits operation in any random orientation. While most of the spacecraft components can be operated satisfactorially from $-20^{o}C$ to $+70^{o}C$, the battery prefers an environment between 0^{o} and $+30^{o}C$. A separate array of solar cells on the base of the spacecraft provides heater power for the experiment in the worst case cold condition.

POWER SUBSYSTEM

The power system consists of four paddles with solar cells on both sides; a 6-ampere-hour 8-cell nickel cadmium battery; and a shunt regulator. Still assuming a random orientation, the power system must guarantee 27 watts average power over the entire orbit, both nighttime and daylight portions. This is particularly important to astronomy experiments, which frequently get their most useful data at night, when scattered sunlight cannot interfere with the sensors. The basic spacecraft functions require 17 watts average power, allowing 10 watts for the experiment. From Figure 4 you can see the variation in available power as a function of sun angle. Total power

CROSS-SECTION
OF
BASIC SPACECRAFT
STRUCTURE

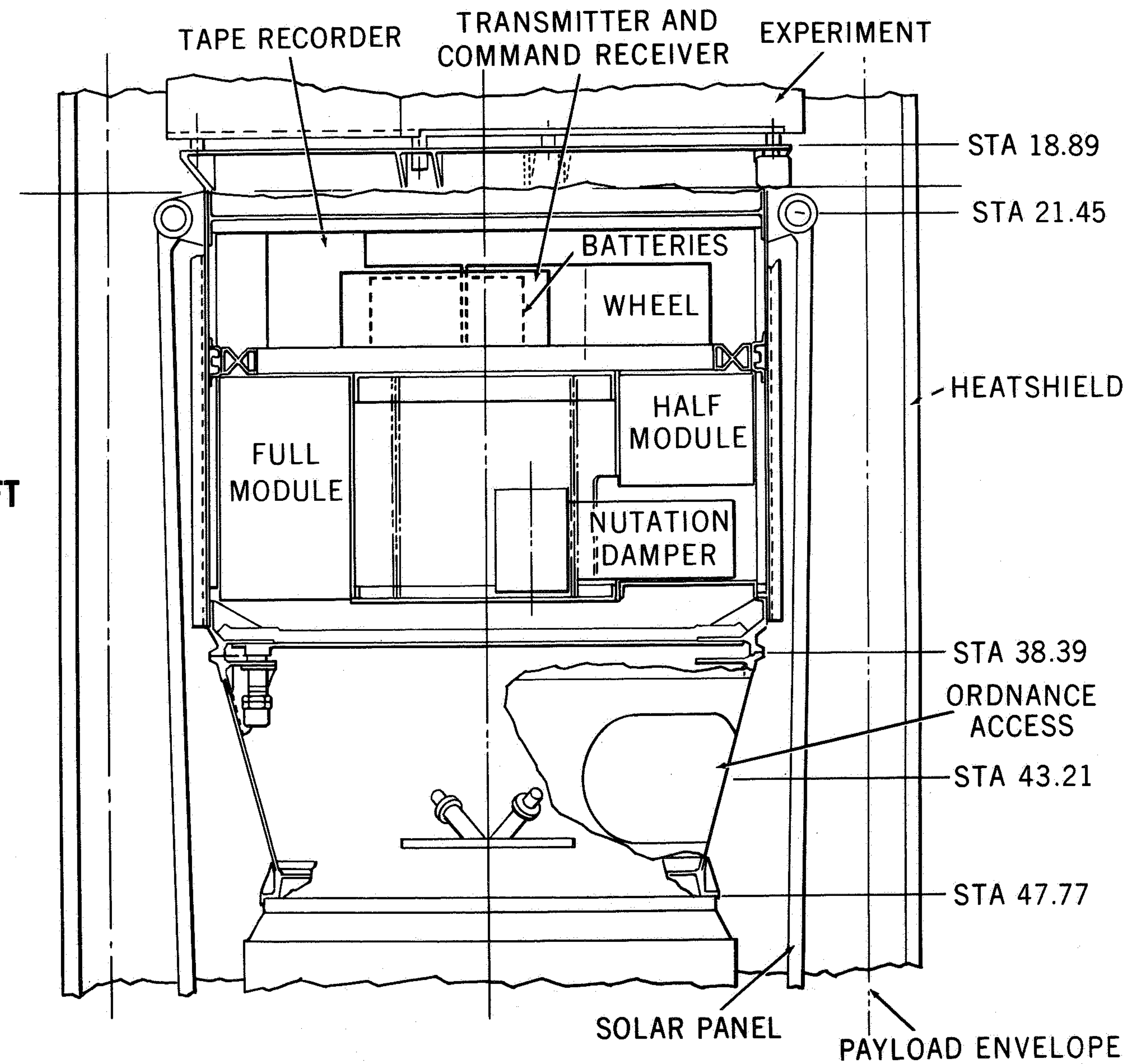


Figure 3

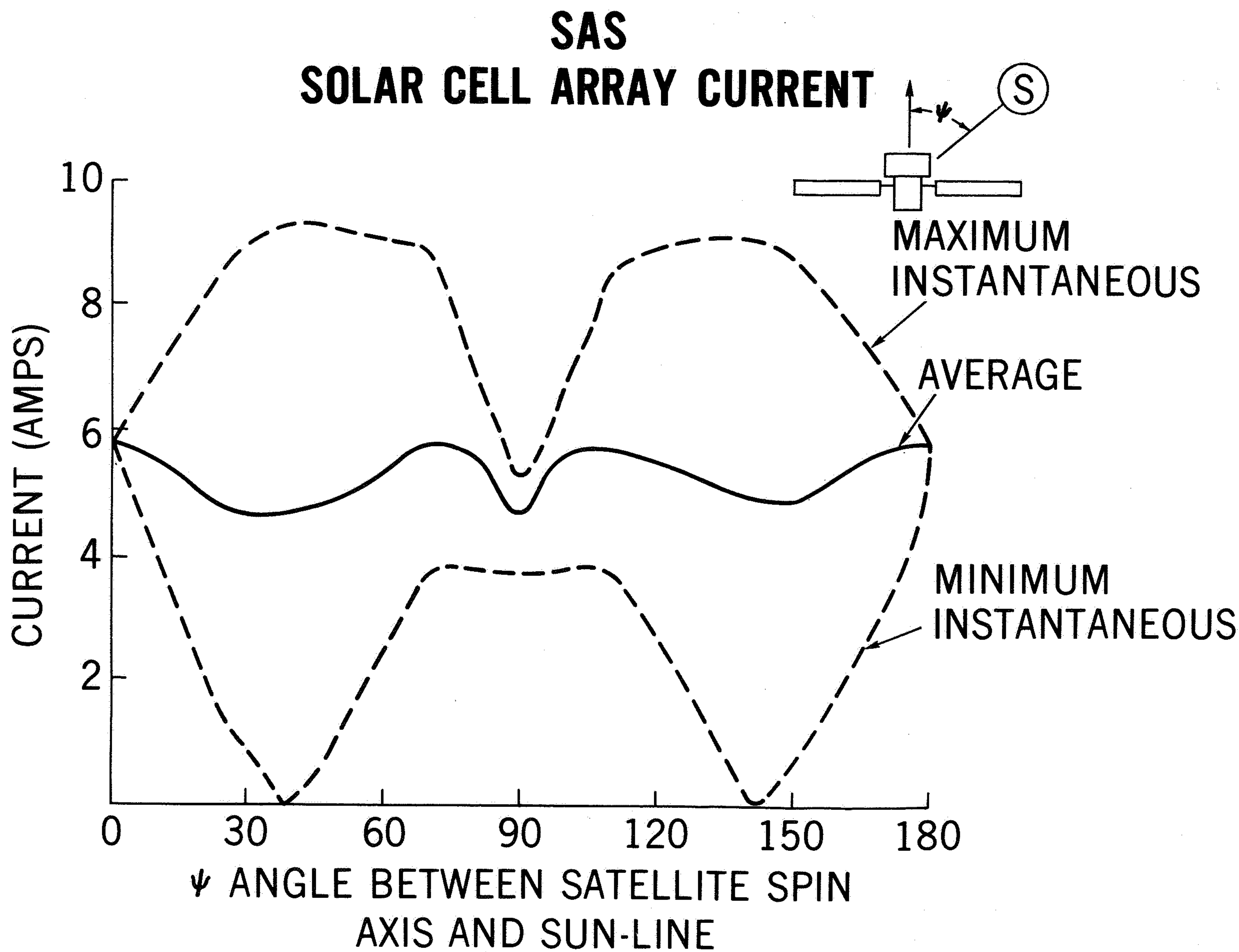

SAS
SOLAR CELL ARRAY CURRENT
S
ψ
MAXIMUM
INSTANTANEOUS
AVERAGE
MINIMUM
INSTANTANEOUS
CURRENT (AMPS)
10
8
6
4
2
0
30
60
90
120
150
180
ψ ANGLE BETWEEN SATELLITE SPIN
AXIS AND SUN-LINE

Figure 4

includes operational needs plus enough additional power to charge the battery to provide the same amount of power through the nighttime portion of the orbit.

A block diagram of the power subsystem is shown in Figure 5. The solar cell array is connected into two separate sections, the main array and an auxiliary array. The auxiliary array is used to provide redundant power for the command subsystem and to trickle charge the battery if it has been removed from the main bus by the low voltage sensing switch.

The six ampere-hour nickel cadmium battery is protected by two redundant charge control devices, a coulometer and an enhanced zener. The coulometer senses the total energy taken from and returned to the battery by integrating the voltage developed across the battery telemetry resistor. When the energy which was taken from the battery on discharge has been returned, the coulometer will reduce the battery charge current to a "safe", preset trickle charge level by shunting the remainder. This unwanted current is shunted internally if the battery is cold, or externally if the battery is hot. The enhanced zener is a voltage limiter. When the battery voltage reaches a preset level, the enhanced zener will actuate a shunt, decreasing the battery charge current. Thus the battery charge current will be reduced to a level which will cause the battery voltage to be equal to the enhanced zener limit voltage. As with the coulometer, the shunted current may be dissipated internally depending on battery temperature. The power subsystem will normally be operated with both the coulometer and the enhanced zener. However, it can operate with either one alone.

The low voltage sensing switch continuously monitors the voltage of the main bus. If it drops below 8.8 volts (normal voltage is 10.7 v.), power is switched from the main converters and the battery in order to protect them from damage due to possible shorted loads. In this "solar only" mode, the battery will be trickle charged by the auxiliary array, while the only loads on the main array are the dual command systems and the heater circuits. Each circuit can be switched out by command if it becomes defective.

SAS POWER SYSTEM

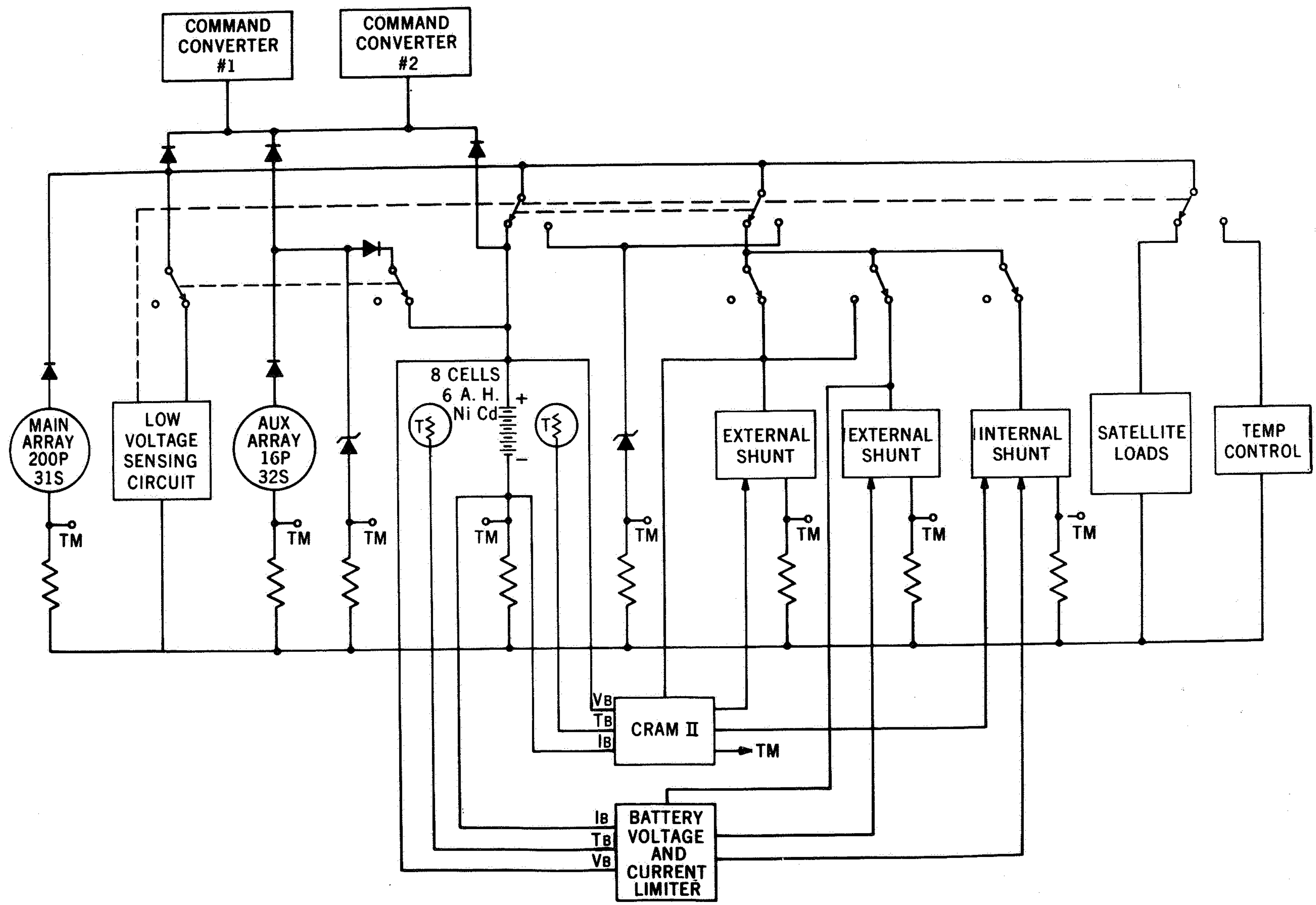


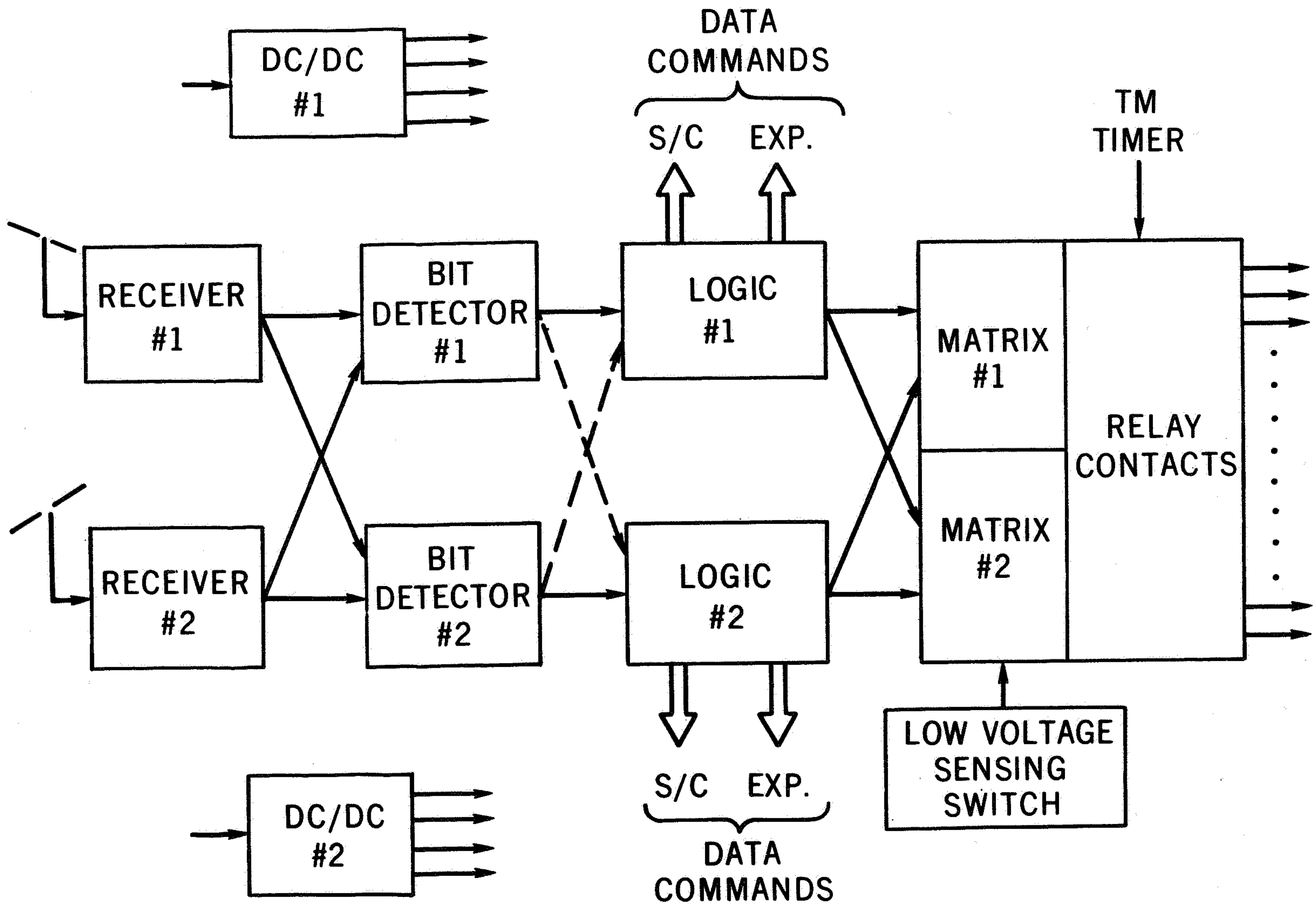
Figure 5

COMMAND SUBSYSTEM

SAS employs the NASA standard PCM command system. The spacecraft carries dual redundant receivers and decoders. The redundancy continues through the relay coils themselves, but a single set of contacts is used. Figure 6 shows the command system in block diagram form. Independent dipole antennas are mounted on the tips of two adjacent solar paddles. The command signal, a pulse code modulated, frequency shift keyed, amplitude modulated (PCM/FSK-AM/AM) signal, is demodulated by either or both command receivers and appears at the video output. Bit synchronization is obtained from a signal that has been amplitude-modulated onto the data subcarrier. The video signal is sent to a command decoder, which tests the tones for authenticity by examining timing and frequencies. If the tones are proper, the decoder converts the tones to digital levels and sends the levels and appropriately timed clocking pulses to the logic circuitry. The decoder also applies power to the standby portions of the logic, beginning with the first correctly received tone burst and remaining on long enough to complete the command. The logic circuitry accumulates the decoded levels in a shift register and, if the satellite address bits are proper, it activates the proper row and column of the relay switching matrix to switch the desired relay or relay cluster.

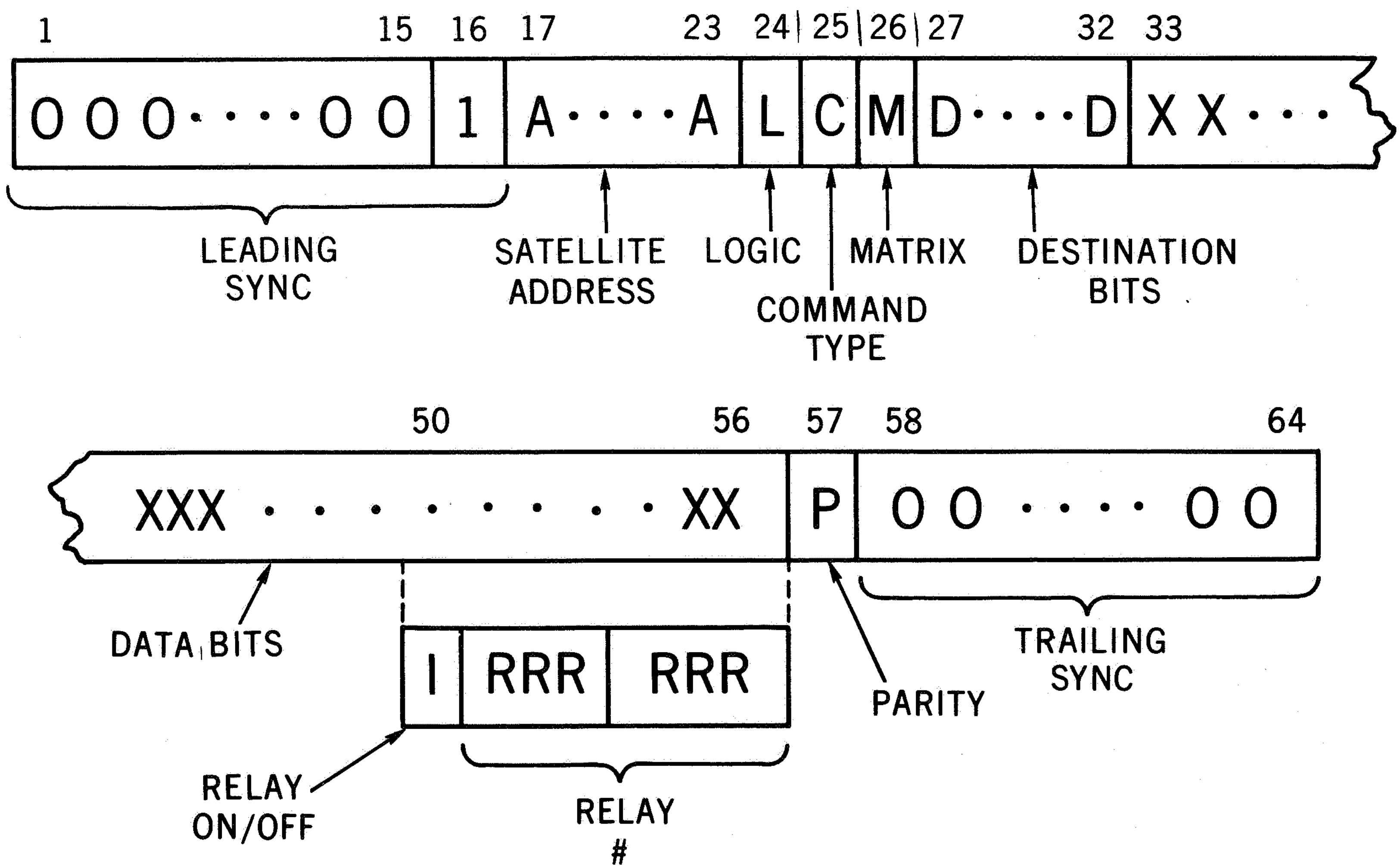
The redundant interconnections show that only one receiver, one command logic, and one relay switching matrix need be operating to send commands. One bit in the command word (Figure 7) addresses the particular logic circuit and one bit the particular relay switching circuit to be used, so that only one logic and one matrix are activated during any given command. Each half system has its own power converter. The system's magnetic latching relays have two coils per relay, one for each system half, and consume power only during a command. The direction of current flow in any given relay coil determines the state in which the relay will latch. Because of these dual coils, actually located in the relay switching matrix, the system's redundancy continues up to the magnetic flux circuit in the relay pole piece, and no single relay coil open or short can disable the system.

The command system provides SAS with 35 on-off relay commands, 25 for the spacecraft and 10 for the experiment. In addition, if bit 25 in the command word specifies a "data command" instead of a "relay command", then bits 33



SAS COMMAND SYSTEM

Figure 6



SAS COMMAND WORD

Figure 7

through 56 are routed, as specified by bits 27 through 32, to a location other than the relay matrix for separate decoding. This permits an experimenter to increase his allocation of 10 on-off commands by decoding up to 2^{24} within the experiment.

TELEMETRY SUBSYSTEM

Due to the low equatorial orbit chosen for SAS, the spacecraft can be seen regularly by only one of NASA's Satellite Tracking and Data Acquisition Network (STADAN) stations. For this reason, the spacecraft telemetry subsystem (Figure 8) must include on-board storage of the data collected during the remaining 90% of the orbit. Therefore, the pulse code modulated/phase modulated (PCM/PM) telemetry system has two basic modes of operation - record and playback. In the normal, or record mode, digital data from various sources are multiplexed and sent to be stored serially in Manchester Code on an endless loop tape recorder and to phase-modulate a VHF transmitter. This continuous output, radiated through a turnstile antenna at the tip of one of the solar paddles, is used as a beacon for tracking the satellite and for the collection of real-time data. As the satellite passes over its data acquisition station, however, the tape recorder is commanded into playback mode and the data from the full 96 minute orbit phase-modulates the same transmitter with a 30 kbps signal. Simultaneously, the transmitter is switched into its high power (2 watt) mode. The tape recorder and the transmitter automatically return to the record mode of operation after completion of the 3.4 minute playback cycle, or upon ground command. In a special operating mode, real-time data can be transmitted at 2 watts. This system meets the requirements of the NASA/GSFC Aerospace Data Systems Standards for PCM telemetry systems. Phase deviation is $\pm 64^{\circ}$ peak, with a realtime bandwidth allocation of $\pm$15kHz and a tape recorder playback allocation of $\pm$45kHz. Linear phase filters, having their 3 db points at 1kHz and 30kHz respectively for record and playback mode, are used for premodulation filtering.

The 1000 bps real-time data rate was determined by optimizing experimenter and housekeeping data requirements with a reasonable length of tape (300 feet) and a reasonably conservative bit packing density for single track recording (1667 bits per inch). The tape recorder, of flight-proven endless loop design, weighs only 7 pounds

SAS TELEMETRY SYSTEM

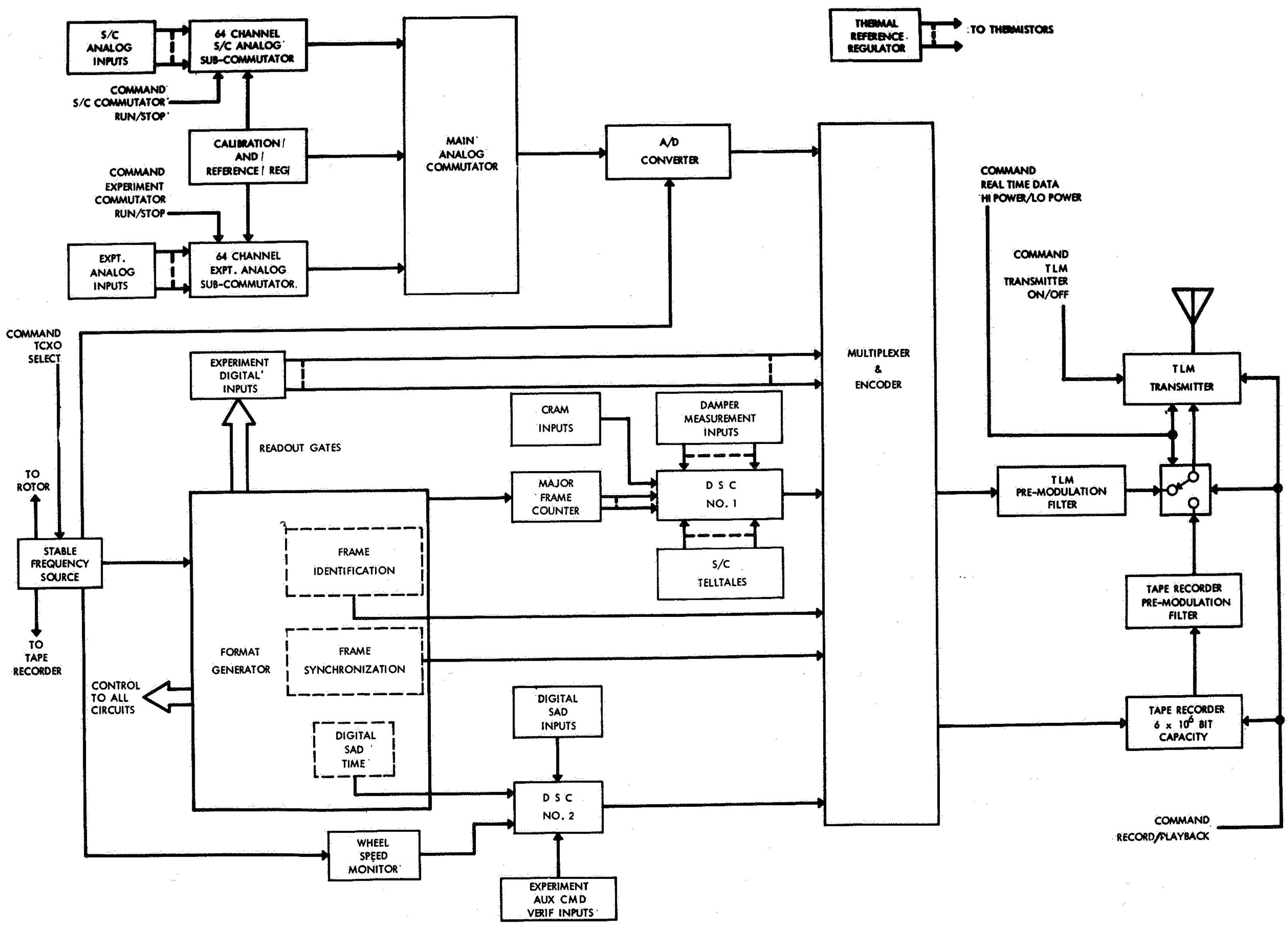


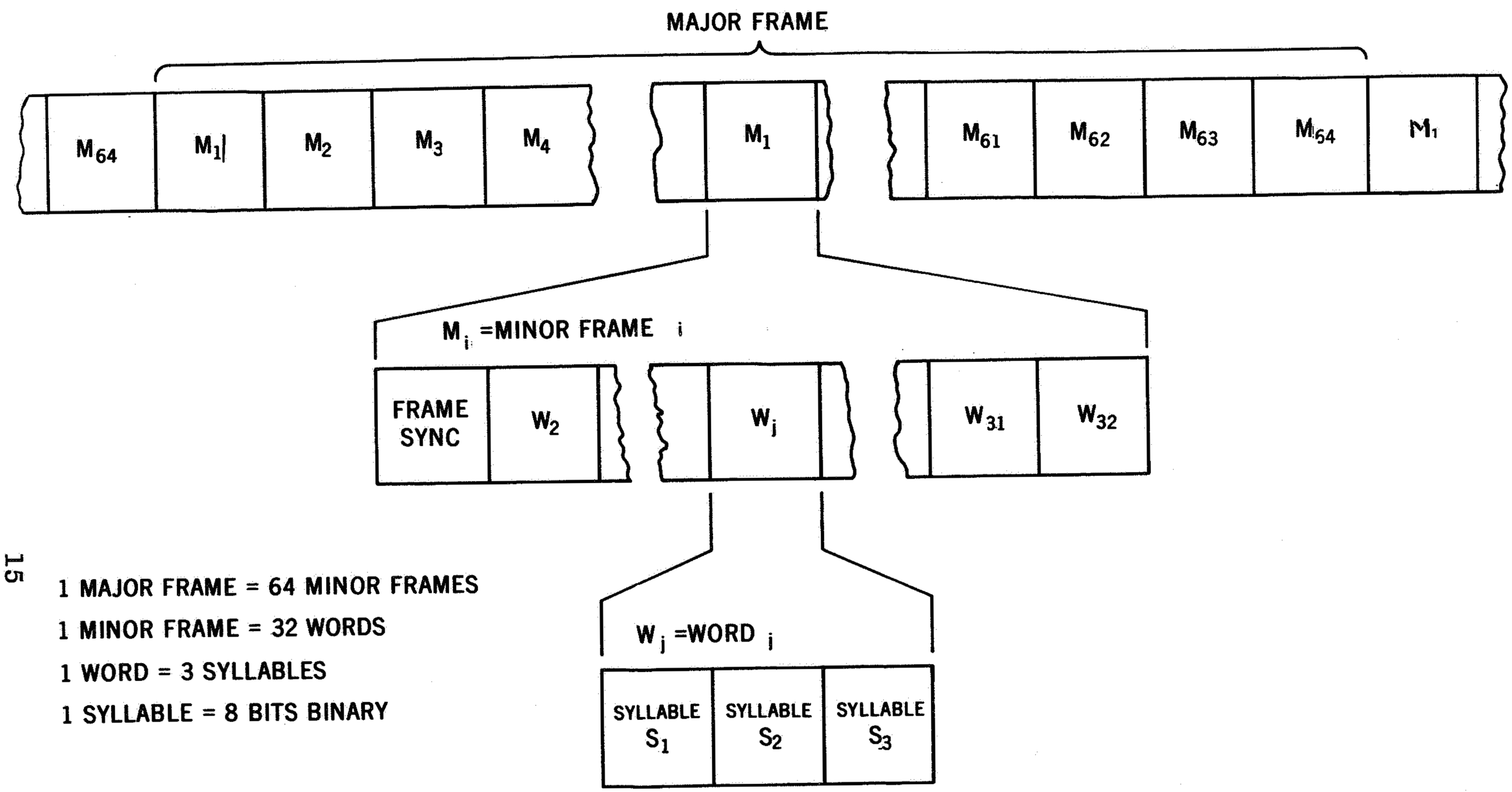
Figure 8

and requires 1.5 watts in record mode and 3.0 watts in playback mode. Wow and flutter will be kept below 2% peak-to-peak, with jitter below 3% peak-to-peak. Commands will bypass the output reclocking circuitry and energize the erase head, as well as provide the obvious record/ playback and on/off control. Digital data recording was selected for its better signal-to-noise properties and the fact that the data could be provided to the telemetry system most easily in digital form.

The basic format is shown in Figure 9. The minor frame is composed of 32 words, each having three 8-bit syllables, which permits easy handling of 8, 16, or 24 bit data words. Of these 96 syllables, the first three are allocated to frame synchronization. The frame synchronization word used is the optimum 24 bit code word determined by Maury and Styles[1], 111110101111001100100000. One syllable is allocated to frame identification, three to 64 position analog subcommutators, two to digital subcommutators, one for a parity word and the remaining 86 for data. The period of the minor frame is 0.768 second and, for the major frame, 64 times that or 49.152 seconds. The basic timing source for the system is a temperature-compensated 1.024MHz crystal oscillator, whose frequency variation is less than one part in 10^6 per orbit. It serves as the source for all readout gates, clock signals to control the multiplexing of spacecraft and experiment data, motor drive frequencies for the tape recorder and rotor, and pulses for the 20 bit accumulator which is incremented by the minor frame identifier every major frame, and sampled every eight minor frames. Unambiguous time correlation for about 20 months is provided.

While most of the data originates in digital form, many of the housekeeping functions are analog and must be converted. A dual-slope integrating analog-to-digital converter, accurate to 8 bits, is used for this purpose. It is clocked at 16kHz and has an aperture of 8 ms. While its normal inputs are 0 to 250mv, the telemetry system can provide a variety of attenuators so that signal levels of ±0.5, ±1.25, ±2.5, ±5.0, or ±7.5 volts can be accepted. An additional feature of the analog subcommutators is that it is possible to command either of them to stop and sample continuously, at the minor frame rate of 0.768

1 Jesse L. Maury, Jr. and Frederick J. Styles, "Development of Optimum Frame Synchronization Codes for Goddard Space Flight Center PCM Telemetry Standards", NTC'64 Proceedings



SAS PCM TELEMETRY SYSTEM FRAME FORMAT

Figure 9

second, any one of its 64 positions. After receiving the RUN command again, it will automatically resynchronize itself at the beginning of the next major frame.

One of the digital subcommutators has 16 channels. Twelve of the channels accept 8-bit parallel data while the other four accept 8-bit serial data. The second digital subcommutator has 8 channels, each of which accepts 8-bit serial data. These digital subcommutators provide us with command verification, the unambiguous time code, a monitor of the rotor speed, and time correlation to 24 ms for the digital solar aspect sensor data.

This telemetry system is less than 0.2 cubic foot, weighs 14 pounds, and consumes 4.5 watts. It is built in modular form for easy expansion, and the design is such that the present hard-wired format could be replaced by an in-flight programmable unit if these requirements occur in the future.

CONTROL SUBSYSTEM

The most unique feature of the Small Astronomy Satellite is its control system. An evolutionary system is planned which will permit the present scanning system to progress to an accurately-pointed, three axis stabilized spacecraft, a most valuable tool in astronomy.

First, a look at the basic control system. The prime contractor for SAS, the Applied Physics Laboratory of the Johns Hopkins University, has a well-established reputation in the field of magnetic control. An obvious advantage of the use of the earth's magnetic field for torquing near-earth satellites is the elimination of the need for expendables, and their inherent life limitations.

Experiments can look along the Z-axis of the SAS spacecraft or perpendicular to it. If one is looking out perpendicular to the Z-axis, and the spacecraft is rotated slowly about the Z-axis, then a swath is swept out of the celestial sphere. The vector made by the Z-axis in space can be moved, and another swath swept. Thus, ultimately, the entire celestial sphere can be scanned.

This sweeping technique is valid even if the Z-axis wobbles, but data reduction can be made much easier if it is stable. One way of achieving stability is by spinning the spacecraft. This is quite satisfactory for many experiments looking along the Z-axis. However, a high spin rate will reduce the effective time-on-target for perpendicular-looking experiments and, consequently, the signal-to-noise ratio of the celestial sources. An alternative method of adding the required angular momentum along the Z-axis is to include a momentum wheel whose axis is parallel to the Z-axis.

Figure 10 shows the various components of the control system. Controlling the Z-axis orientation is accomplished by energizing a torquing coil. This electromagnet, acting like a compass needle, attempts to align the spacecraft with the earth's magnetic field. At the maximum rate, using a magnetic dipole of 5×10^4 pole-centimeters, it is possible to move the Z-axis of the spacecraft 1.72 degrees per minute.

Figure 11 shows how a maneuver might be performed to point the Z-axis to a new direction as requested by an experimenter. This particular sequence is designed to result in no change in right ascension, but only in declination. In normal operation a combined motion in both right ascension and declination will be accomplished with the aid of a computer. The computer must store the values of the earth's magnetic field in each predicted satellite position. Knowing this, the dipole moment of the satellite, and the desired change of position, the computer will determine at what point in the orbit the torquing coil is to be turned on and for how long. Time delays of up to an hour are available in the satellite so that the maneuver can begin at the optimum time in the orbit, not just over the command station. Time increments for this delayed maneuver are 2.5 minutes, both for the delay and for the maneuvering period. Finer control can be obtained by turning the torquing command on and off while within sight of the station. Once the position is achieved, it is held primarily due to the momentum generated by the rotor, (Figure 12), which is 2.12×10^7 gm-cm^2/sec. Drifting from this position is caused by gravity gradient, aerodynamic, and magnetic torques, the latter resulting from uncompensated dipoles. The total of these drift rates should not exceed 5 degrees per day on the average. To guarantee that the residual magnetism of the spacecraft is minimized, it is provided with a degaussing coil and a chargeable trim magnet system to compensate in orbit for residual magnetism along each axis. It is possible to use this system to produce a specific dipole moment for controlled spin axis drift.

SAS-A SPIN AXIS ORIENTATION CONTROL SYSTEM

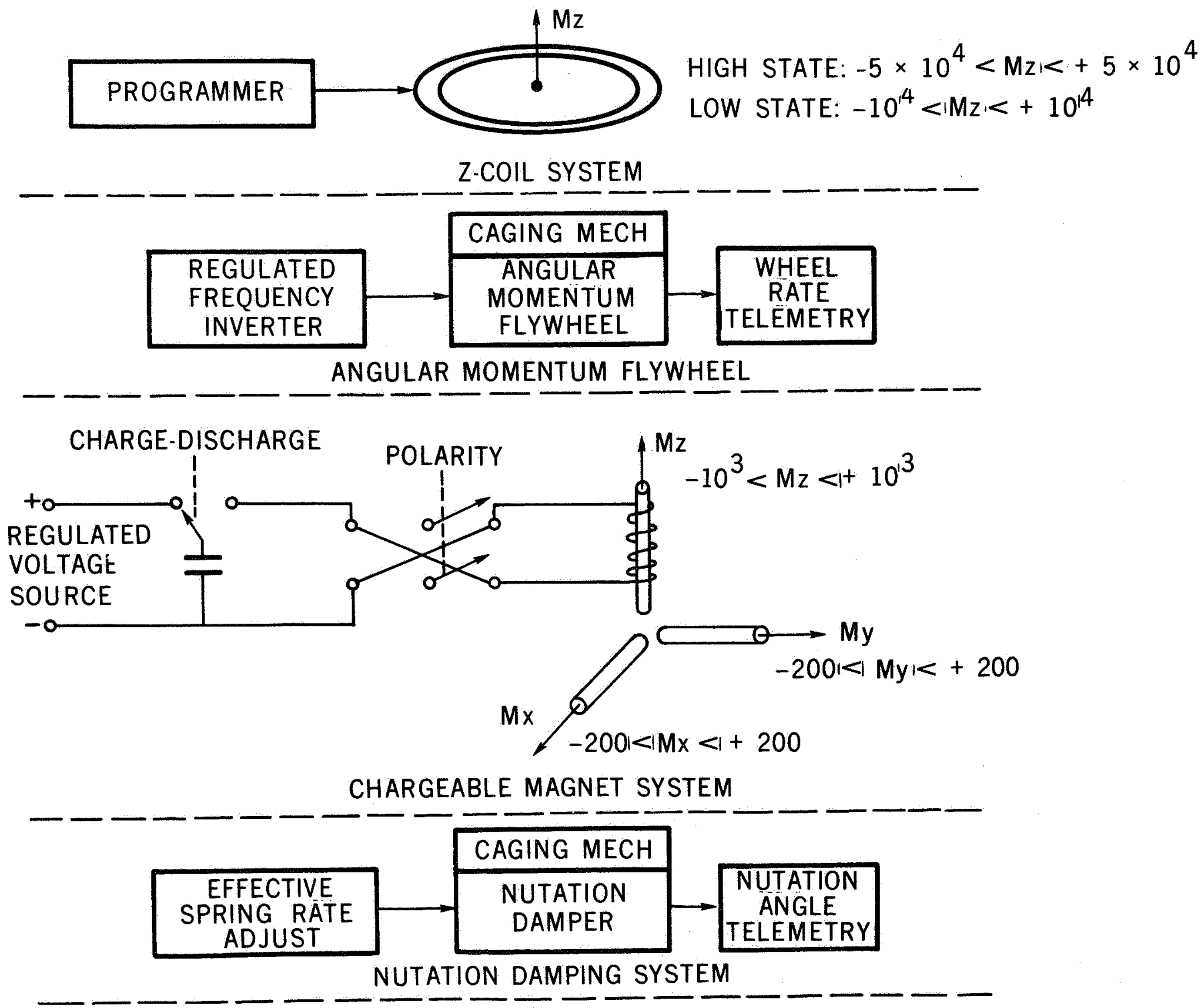


Figure 10

SAS-A DECLINATION MANEUVER SEQUENCE

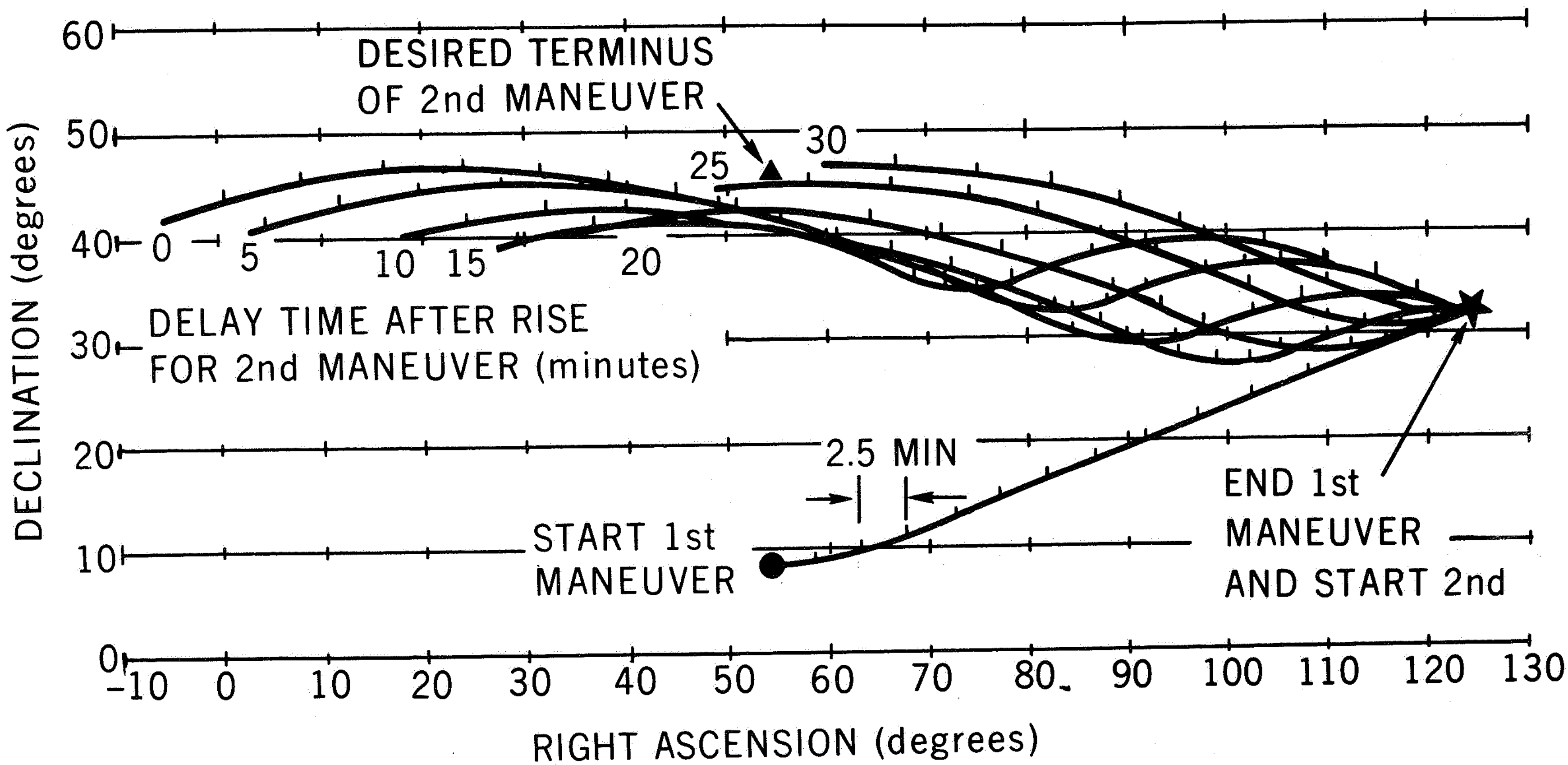


Figure 11

SIMPLIFIED DRAWING OF ANGULAR MOMENTUM FLYWHEEL

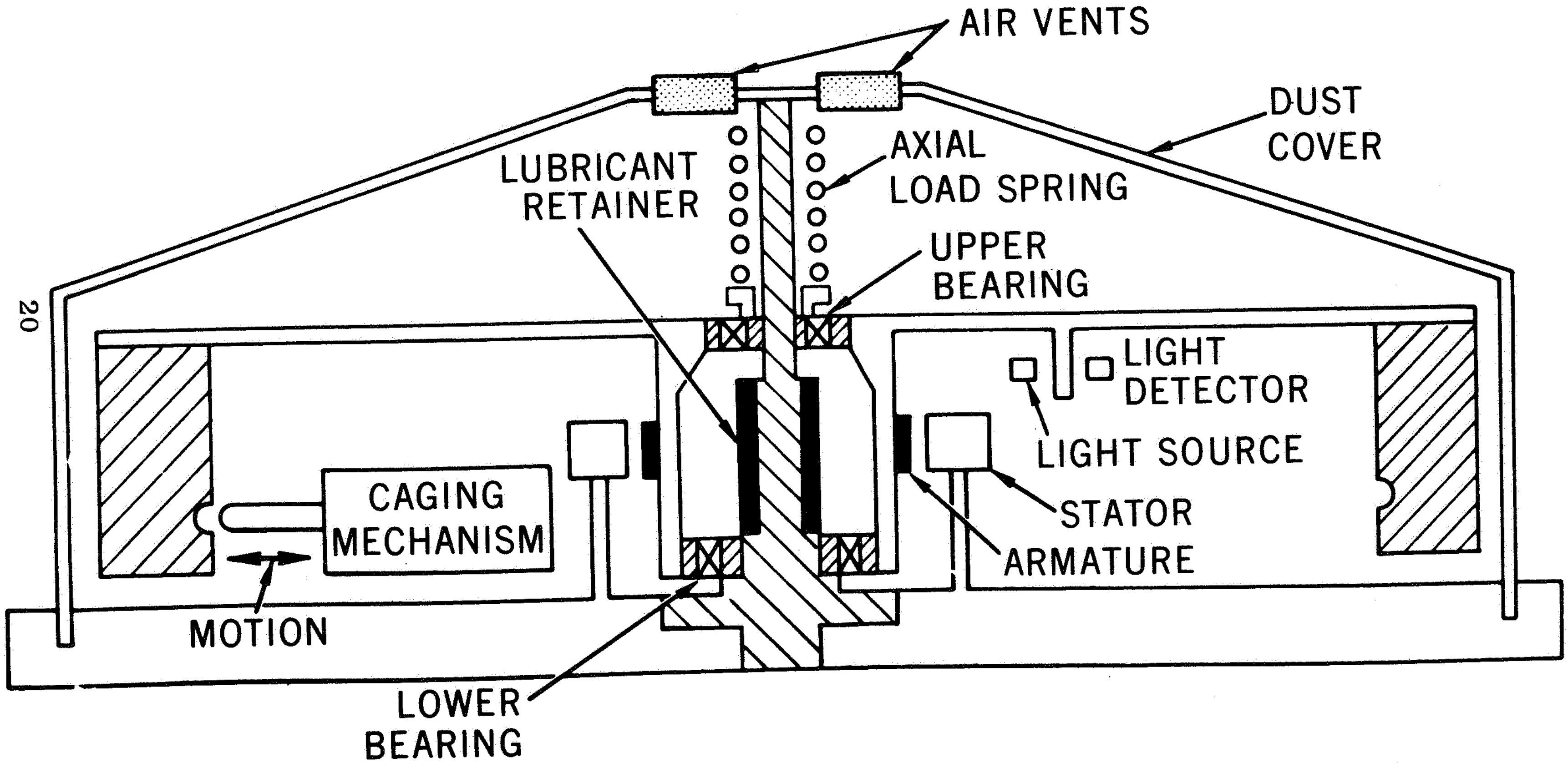


Figure 12

The nutation damper is used to dissipate the lateral components of satellite angular velocity created by the magnetic torquing. It consists (Figure 13) of a torsion wire suspended arm with an end mass and a copper damping vane. If nutations occur, the arm oscillates causing the copper vane to swing back and forth through the gap in a permanent magnet, thus inducing eddy currents in the copper vane. With the rotor on, the residual nutation should be less than 0.2 degree; with the rotor off, it may be a few degrees. The motion of the nutation damper, detected optically by a coded digital mask mounted on the movable arm, is telemetered every 8 seconds via a 7 bit binary word.

The spin rate control system is shown in Figure 14. Signals sensed by the X- and Y-axis magnetometers are amplified and applied in quadrature to the Y- and X-axis coils. This system, in conjunction with the earth's magnetic field, operates essentially as a motor to increase or decrease the speed of rotation about the Z-axis. The maximum rate of change of spin rate is 10 rpm per day.

This summarizes the basic SAS control system with which many astronomy experiments can be performed well. It is ideally suited for sky surveys when used in conjunction with a good attitude determination system. Data telemetered from the magnetometers and a digital sun sensor on board the spacecraft can be used to determine attitude to $\pm 5^{\circ}$ or better.

EXPERIMENTS AND FUTURE DEVELOPMENT

On SAS-A, the principal investigator, Dr. Riccardo Giacconi of American Science and Engineering, Inc., is flying star and sun sensors accurately aligned with the experiment X-ray collimators so that attitude may be determined to a minute of arc. This experiment will provide us with an accurate map of X-ray sources throughout the celestial sphere. After such a survey which will determine source strength, position, and spectral composition, it is obvious that later X-ray experiments will require more accurate pointing capability for detailed study of individual sources. The SAS design permits building on this basic control system by adding a star tracker or star camera system, to provide three axis stabilization to hold the attitude of the satellite to one

SIMPLIFIED DRAWING OF SAS-A NUTATION DAMPER

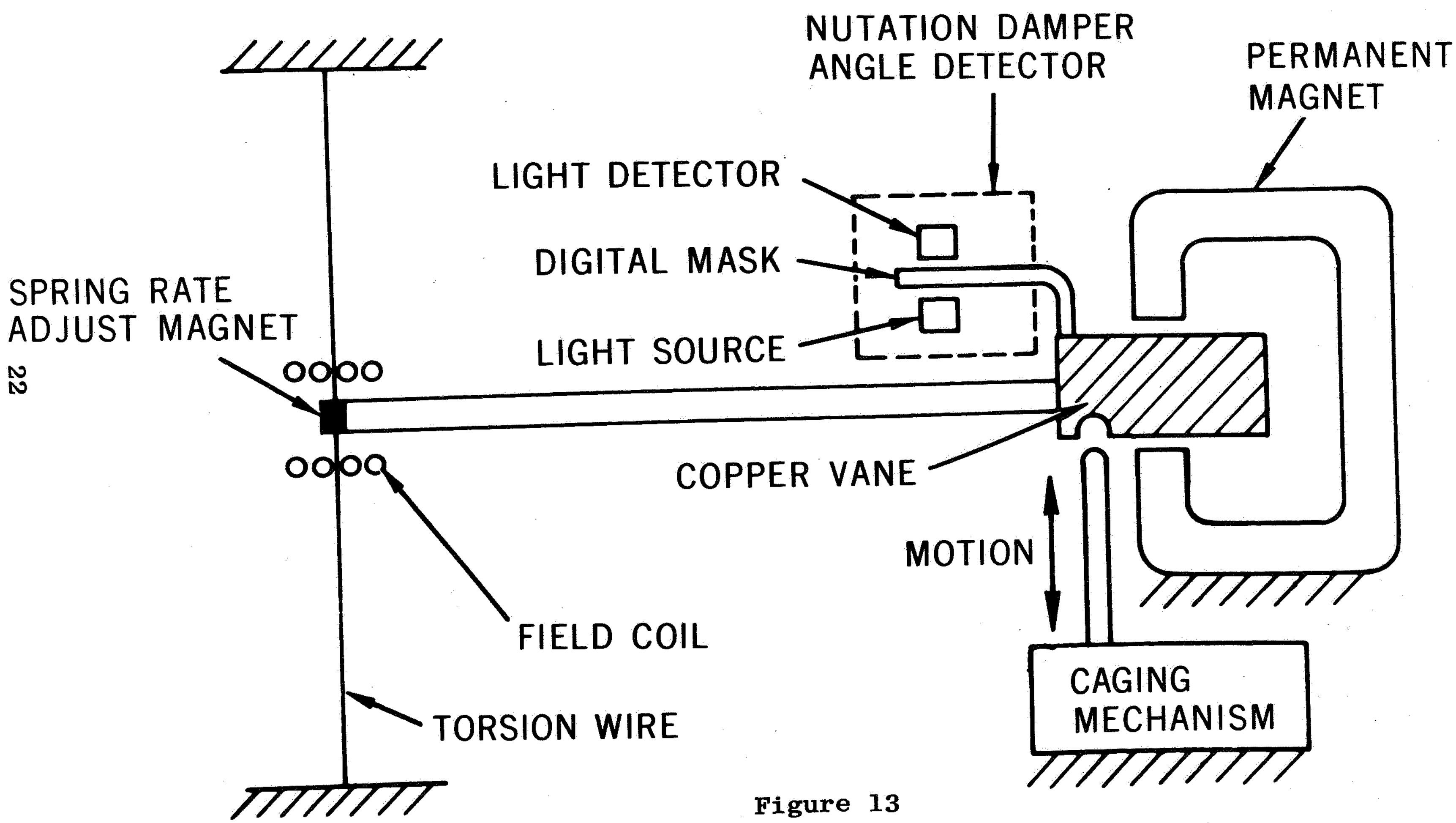


Figure 13

BLOCK DIAGRAM OF SPIN CONTROL SYSTEM

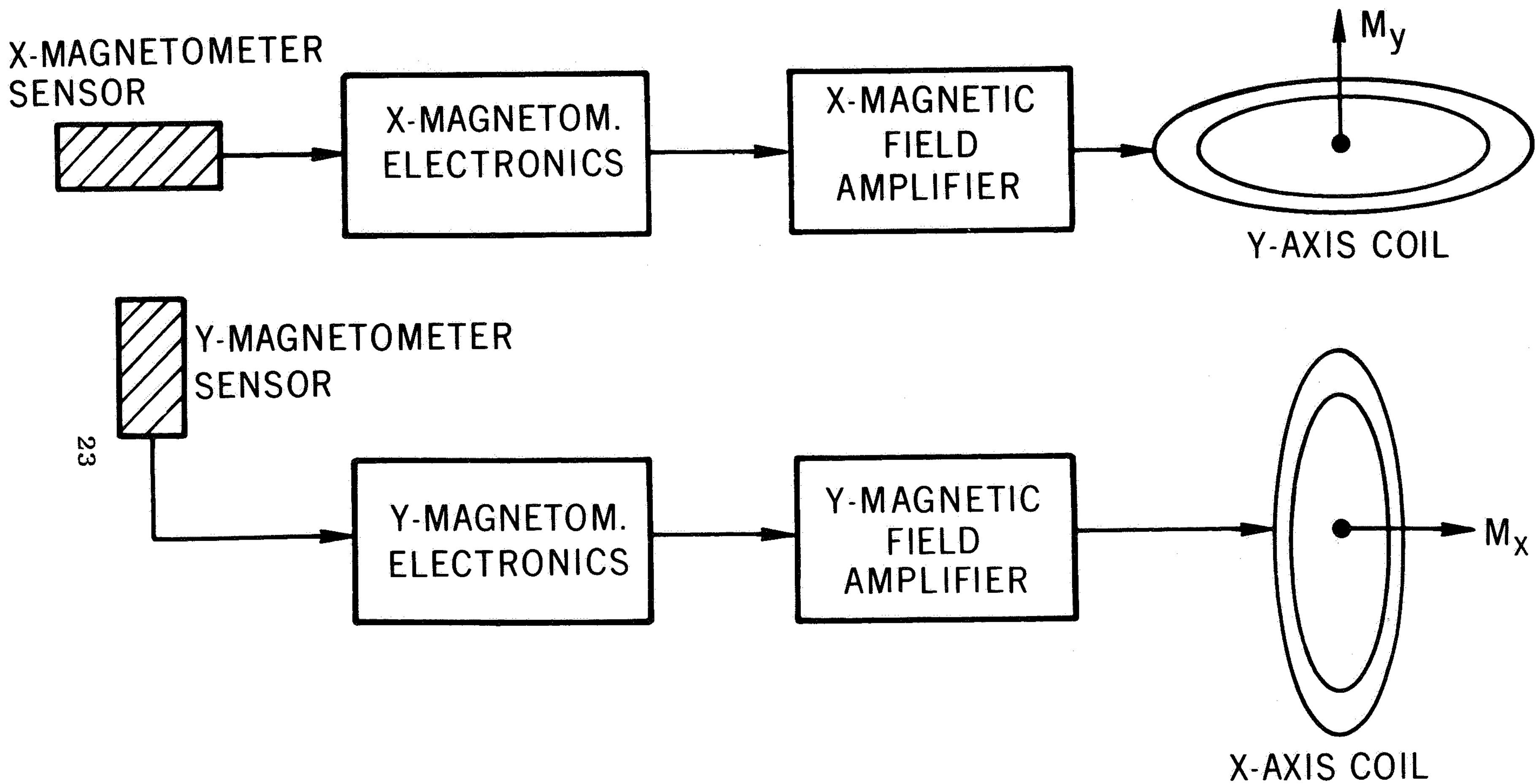


Figure 14

minute of arc or better. This would permit useful experiments in the UV and IR regions of the spectrum as well.

Rough positioning of the spacecraft would be accomplished by the basic control system as described above, and the exact attitude determined by ground computer. Then, for example, a program could be transmitted for storage in the spacecraft defining a pattern that the star camera should see and fine torquing would be used to find and maintain the proper position. Two sets of optics with about 10° fields of view located at 90° to each other could permit three-axis attitude detection to a resolution of 10 seconds of arc, and control to 1 minute of arc. Error signals would control the speed of the basic rotor whose excess momentum would be dumped into the earth's magnetic field - again, no expendables. Small gyros or a worm gear tilting mechanism would be suitable for fine control of the X- and Y-axes. A system of this general type is planned for SAS-C and later flights. SAS-B will be a sky survey of Gamma-Ray sources, with Dr. Carl Fichtel of the Goddard Space Flight Center as principal investigator. The basic control system is adequate for SAS-B as the Gamma Ray Spark Chamber has a 45° field of view and does not require very fine pointing.

SUMMARY

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SUMMARY OF SAS CHARACTERISTICS

LAUNCH VEHICLE

Scout with FW-4 fourth stage

ORBIT

Altitude: 300 n. mi. (555 km) $\pm$40 n. mi. (74 km) (one sigma)
Inclination: 2.9° $\pm$0.15° (one sigma)
Period: 95.8 minutes
Eccentricity: 0.008

SIZE

Spacecraft: Shape - Cylinder
Diameter - 22"
Height - approx. 20"
Volume - approx. 4.4 cu. ft.

Experiment: Shape - Cylinder topped by truncated cone (See Fig. 15).
Base diameter - 30"
Height of cylindrical portion - approx. 14" or 29"
Height of truncated cone - 15.2"
Top diameter - 17.7"
Volume - approx. 6.4 cu. ft. or 12.5 cu. ft. (for the "pointed SAS" about 1 cu. ft. must be made available from the experiment volume)

WEIGHT

Spacecraft: 180 pounds

Experiment: 150 pounds - must have low C.G. - See Figure 16.
(130 pounds on "pointed SAS")

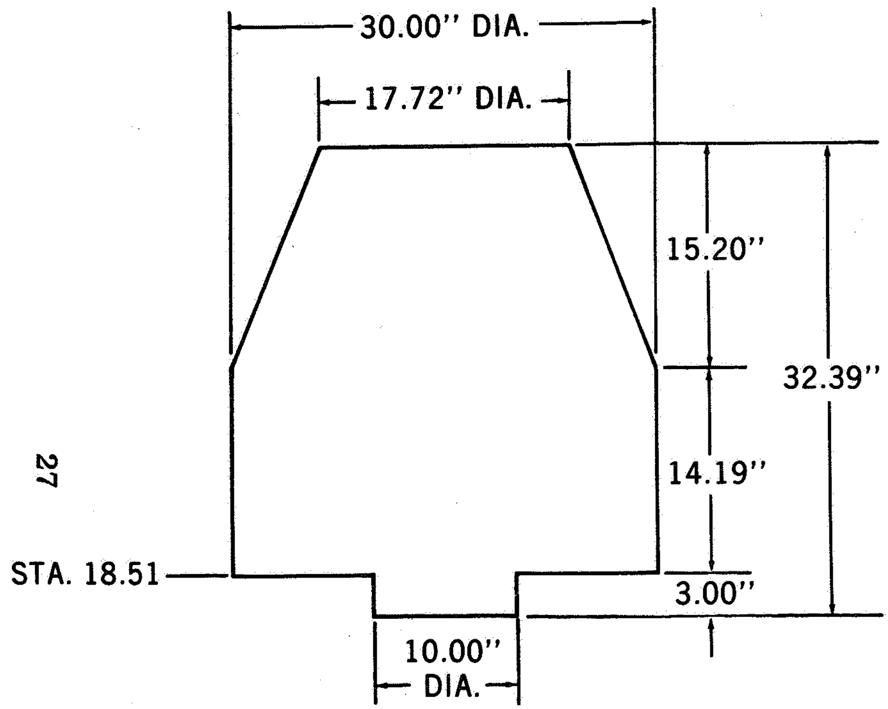
CONTINUOUS AVERAGE POWER

Spacecraft: 17 watts

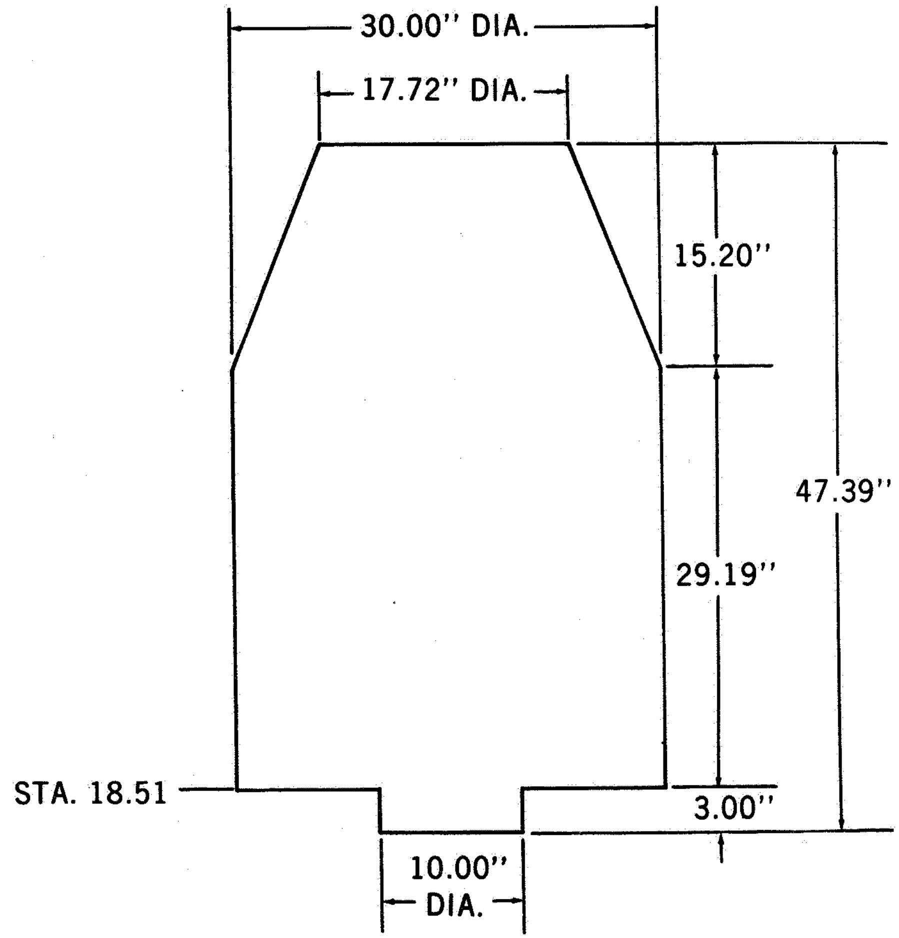
Experiment: 10 watts (9 watts on "pointed SAS")

Voltage: +10.7 v.d.c. $^{+10\%}_{-15\%}$

Ground: Three-wire system - separate chassis, signal and power grounds



STANDARD SCOUT SHROUD

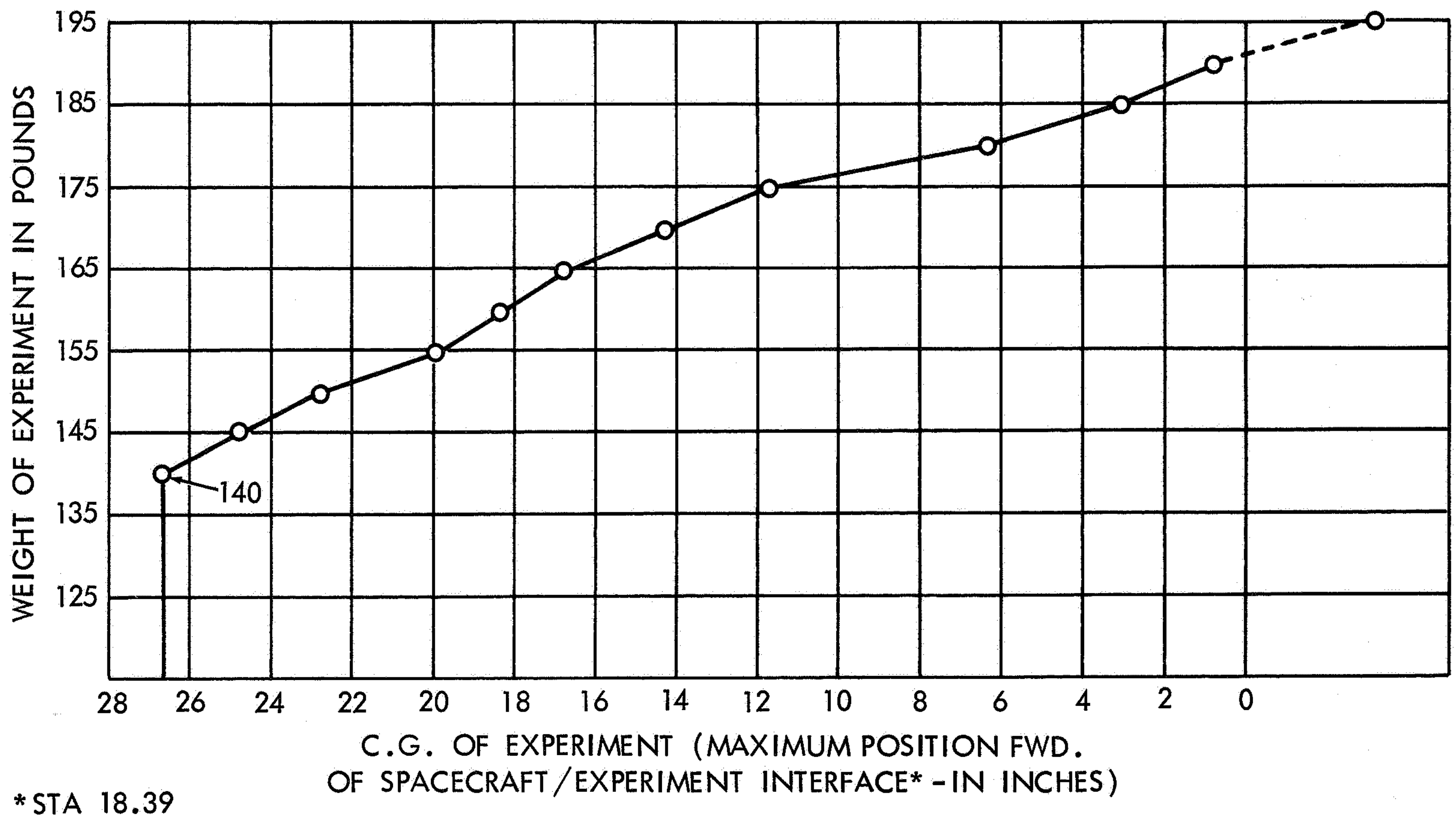


ELONGATED SCOUT SHROUD

OUTLINE DRAWING OF
SAS EXPERIMENT PACKAGE

Figure 15

SMALL ASTRONOMY SATELLITE EXPERIMENT WT. VS C.G.



*STA 18.39

Figure 16

TELEMETRY

Type: PCM split phase
Real Time and Record Rate: 1000 bits per second for 96 minutes (includes experiment and housekeeping data)

Note: 16,000 bits at 500 kbps or less could be accepted instantaneously into a buffer for readout onto tape at 1000 bps.

Storage Method: Endless loop tape recorder
Capacity: 5.4×10^6 bits
Playback Rate: 30 kbps for 3.4 minutes
Transmitter: 136 MHz (90 kHz bandwidth)
Encoder: 8 bit analog to digital converter will provide better than 1.0% accuracy for signals up to 1 cps
Input Signal Levels:

Analog: ± 7.5v., ± 5.0v., ± 2.5v., ± 1.25v., or ± 0.5v. with source impedance of 5kohms.

Digital: +4v. for a "1" and +0.3v. for a "0" into a standard LPDTμL

TRACKING

Minitrack
Orbital Accuracy: 10 km

COMMANDS

NASA STADAN PCM System
Spacecraft: 50 commands (25 "on" and 25 "off" commands)
Experiment: 20 commands (10 "on" and 10 "off" commands" (can be extended up to 2^{24} by using logic internal to the experiment)

CONTROL SYSTEM

Basic System:

Positioning of Z Axis - $<1^{\circ}$ relative to known position

Average Drift of Z Axis - 0.2 arc-min. per minute (5 degrees/day)

Spin Rate - 0-60 rpm

Attitude Measurement - $\pm 3^{\circ}$ (if greater accuracy is needed, additional sensors may be provided by the spacecraft or as part of the experiment)

Pointed SAS:

Three-axis stabilization to 1 arc-minute

ENVIRONMENT

Temperature Range: $-20^{o}C$ to $+50^{o}C$

Humidity: Must survive conditions of 100% humidity

Vibration: Table I gives the qualification level vibration specifications at the spacecraft-experiment interface. These are derived from data from the Scout vehicle and are intended to meet the requirements of GSFC Document S-320-S-1, "General Environmental Test Specification for Spacecraft and Components Using Launch Environments Dictated by Scout FW-4 and Scout X-258 Launch Vehicles" dated May 20, 1966.

Magnetism: Residual magnetic moment of experiment should be kept below 500 pole-cm.

TABLE I

QUALIFICATION LEVEL VIBRATION SPECIFICATIONS

RANDOM - 4 MINUTES EACH AXIS

	Frequency (Hz)	Power Spectral Density (g^2/Hz)
Thrust Axis	20-50	0.1
	50-100	Increases at 10 db per octave
	100-150	1.0
	150-370	Decreases at 10 db per octave
	370-2000	0.07
Lateral Axes	20-40	Increases at 10 db per octave
	40-150	1.0
	150-370	Decreases at 10 db per octave
	370-2000	0.07

SINUSOIDAL - 2 OCTAVES PER MINUTE

	Frequency (Hz)	Acceleration
Thrust Axis	20-95	0.187 inches Double Amplitude
	95-175	40
	175-250	30
	250-400	15
	400-2000	7.5
Lateral Axes	15-36	0.13 inch Double Amplitude
	36-200	8
	200-400	5
	400-2000	7.5

SPECIAL SAS REQUIREMENTS

In designing experiments to be flown on the Small Astronomy Satellite (SAS), in general the requirements of NPC 200-2 "Quality Program Provisions for Space System Contractors" dated April 1962 must be met. In addition, parts and materials will conform to the current GSFC Preferred Parts List. While specific details have to be worked out for each experiment, this list is provided for guidance.

PARTS SCREENING

In general, parts screening will be as described in the current GSFC Preferred Parts List. Only hi-rel parts will be used. Specifically, for all semi-conductors, i.e., diodes, transistors, and integrated circuits, the following screening sequence is required as a minimum:

a. Visual inspection before sealing

b. Temperature cycling -65°C to maximum rated storage temperature

c. Centrifuge

d. Electrical test with variables data recorded

e. 336 (±36) hours burn-in at 100°C and 80% of part rated power

f. Electrical test with variables data recorded. Parts will be rejected if outside of acceptable variables limits.

g. Fine and gross leak tests

h. Final inspection

All parts must be approved by GSFC before they can be used. Lists must contain the following information as a minimum:

a. Type of component

b. Value and rating

c. Manufacturer

d. Manufacturer's type and model

e. Manufacturer's screening process specifications to which part is being bought

f. Maximum anticipated electrical stress level

REVIEWS

A design review, a prototype readiness review, and a flight unit readiness review will be held on each experiment and the spacecraft. Malfunction reporting and configuration control will be implemented after design review and prior to the beginning of prototype fabrication.

RELIABILITY ASSESSMENT

A reliability assessment of the experiment will be performed by the experiment contractor.

NOISE AND TRANSIENT PROTECTION

If the experiment requires the use of any high voltages, the experimenter must provide protection for the experiment and the spacecraft systems against high voltage surges or radiated pulses. The experiment must be designed to be insensitive to noise, e.g., that generated by clock pulses or dc/dc converter transients, and is responsible for the elimination of unnecessary transients and noise from his lines by means of suitable filters, shielding, or other means.

Open and Short Circuit Outputs: If opening the TM output would develop a large voltage surge to the multiplexer input (greater than a factor of two over nominal), the output shall be paralleled with protective circuits (Zener diode). Isolation shall be provided from the subsystem monitoring circuits in case of telemetry shorting.

Power Line Transients: Current transients in the main power line to the experiment must not exceed a current-time profile of:

Current	Time
4a.	0-5 μs
1.75a.	5 μs - 0.1 sec.
(normal) 1.0a	after 0.1 sec.